SMART-1: WITH SOLAR ELECTRIC PROPULSION TO THE MOON

Johan SCHOENMAEKERS
Daniel HORAS
Juan A. PULIDO

ESA/ESOC
Robert-Bosch-Str. 5, D-64293 Darmstadt, GERMANY
johan.schoenmaekers@esa.int
daniel.horas.fernandez@esa.int
juan.pulido@esa.int

ABSTRACT *- The SMART-1 S/C will be launched in GTO as additional passenger on Ariane 5. After the lunar transfer phase, it will be injected into a polar orbit around the moon, in which it will stay for the entire six months operational period. The orbit control is done with "Solar Electric Propulsion" only. This is achieved in a fuel efficient way using the moon's gravity in the form of lunar resonances and close lunar swing-by's. After presenting the basic principles and the results of the trajectory design the paper addresses some special issues like eclipse avoidance strategies and navigation at moon approach.*

KEYWORDS: Solar electric propulsion, Moon resonances, Moon swing-by's, Eclipses

INTRODUCTION

SMART-1 is the first of the "Small Missions for Advanced Research in Technology" which have been introduced into the ESA scientific program. The prime objective of SMART-1 is to demonstrate the "Solar Electric Propulsion" (S.E.P.) concept as prime propulsion and as a key technology for scientific deep space missions.

After several trade-off studies, considering budgetary and mass constraints, a lunar mission has been selected. The S/C will be launched in GTO as additional passenger on Ariane-5, which limits the launch mass to 350 kg and implies that SMART-1 has to accept a midnight launch at whatever day between Nov. 2002 and Nov. 2003. The S/C will be transferred into a polar orbit around the moon. Several target orbits are under consideration. For the trajectory design we have assumed a 1000 x 10000 km orbit with pericentre above the south pole. The S/C will remain in this operational orbit for 6 months, after which the earth perturbation will make it impact onto the moon surface. The propulsion will be realised with one PPS-1350 Hall-plasma thruster, providing a force of about 73 mN at an exhaust velocity of 16,4 km/s.

The key to a fuel efficient transfer is the inclusion of coast arcs near apogee when pumping up the orbit to the moon altitude. As a consequence, the perigee will remain low which leads to a high relative velocity when reaching the moon (~ 700 m/s). Because of the low thrust force, moon capture can only be achieved with a low arrival velocity relative to the moon (< 200 m/s). The reduction of the arrival velocity to this value is achieved by a sequence of moon resonances (far moon encounters) and moon swing-by's (close moon encounters).

TRAJECTORY DESIGN

The objective of the trajectory design is to find a numerically integrated and continuous trajectory starting in GTO and ending at the operational orbit around the moon and this in a fuel efficient way and for a given transfer duration which has been selected to be 17 months.

The building blocks of such a trajectory are thrust arcs, moon resonances, moon swing-by's and the lunar capture including the subsequent transfer to the operational orbit around the moon. First the building blocks are presented. Later on it is explained how the building blocks are put together to construct the trajectories.

Thrust Arcs

The start and the end of the thrust arcs is defined in terms of true anomaly. Unless stated otherwise, they are symmetric around pericentre or apocentre.

A thrust arc can be tangential. In this case the thrust direction is along or opposite to the velocity. This will modify the apocentre radius when applied near pericentre and modify the pericentre radius when applied near apocentre.

A thrust arc can also be direction modulated. In this case the direction is tuned to simultaneously change the semi-major axis (a) and change the elevation of the apocentre (β) above the moon orbit plane. The thrust direction at the current point on the orbit follows from:

$$\vec{A} = W_a \cdot \frac{\partial a}{\partial \vec{V}} + W_\beta \cdot \frac{\partial \beta}{\partial \vec{V}} \qquad \text{(EQ 1)}$$

where:

$\vec{V}$ is the velocity at the current point on the orbit
W_a is the weighting factor for the semi-major axis change
W_β is the weighting factor for the apocentre elevation change

The partial derivatives, which are of dimension 3, provide the locally optimal directions to perform each of the changes at the current point on the orbit. The norm of the partials is a measure for the efficiency with which the changes can be made. For the semi-major axis change, the direction is obviously along the velocity. The efficiency is larger near pericentre than near apocentre.

The vector $\vec{A}$ is the weighted sum of the partial derivatives. Its direction is expected to be a good compromise between the local optimum directions and is therefore taken as thrust direction. It is a mean thrust direction taking into account not only the weighting factors of the two changes, but also their efficiency. The magnitude of $\vec{A}$ is a measure for the efficiency of the combined change.

If only the weighting factor for the semi-major axis change is different form zero, the modulated thrust arc reduces to the tangential thrust arc.

Moon Resonances

When pumping up the orbit to the moon altitude, the apogee radius gradually increases. From 200,000 km onwards, the moon starts to significantly perturb the orbit once every lunar revolution, i.e. each 27.4 days. These perturbations are called moon resonances and occur near apogee when the earth-S/C direction is close to the earth-moon direction.

The perturbing acceleration is the difference between the gravitational acceleration induced by the moon on the S/C and on the earth. At the considered distances, the acceleration on the earth is much smaller than the acceleration on the S/C. The moon perturbation is only significant over a rather small part of the orbit near the point of closest approach which is near to apogee. To get a qualitative understanding, one can therefore consider the moon perturbation as an impulsive acceleration acting at apogee in the S/C-moon direction. The magnitude of the perturbation increases with decreasing distance to the moon.

The only parameter of a moon resonance which can be easily controlled is the phasing with the moon when reaching the apogee. It is controlled by tuning the orbital periods prior to the resonance via the length of the thrust arcs.

The perturbing acceleration can be decomposed in an along track (along the velocity), cross track (in the orbit plane perpendicular to the velocity) and an out of plane (perpendicular to the orbit plane) component.

To get an along track component, the projection of the earth-moon direction on the S/C orbit when the S/C reaches the apogee may not be on the line of apsides. If the moon is ahead of the S/C, it is in the direction of the velocity and mainly increases the perigee radius. It is opposite to the velocity and decreases the perigee radius if the moon is lagging behind the S/C.

To get an out of plane component, the moon must be outside the orbit plane of the S/C when the S/C reaches the apogee. It rotates the orbit plane around a line close to the line of apsides, in CW direction if the moon is above the S/C orbit plane and in CCW direction if it is below the plane.

The cross track component is always existing when the S/C is below the moon altitude and pointing away from the earth. Its effect is a slight CCW rotation of the lines of apsides around the orbit normal.

Some small numerical experiments have been performed to support the qualitative explanation and to get an idea about the magnitude of the changes. The results are presented in table 1 and table 2. The phase w.r.t. the moon is defined as the sum of the angle from the earth-moon direction when the S/C reaches apogee to the ascending node of the S/C orbit in the moon orbit plane and the angle from this ascending node to the apogee. For a negative value the S/C is lagging behind the moon, whereas for a positive value the S/C is ahead of the moon. The orbit plane rotation is positive in CCW direction around the line of apsides. The rotation of the line of apsides is positive in CCW direction around the orbit normal.

One sees that a resonance is an ideal means to increase the perigee radius, which remained to low due to the inclusion of coast arcs near apogee. For this purpose, we must tune the orbital periods such that at the resonance the moon is ahead of the S/C by typically 15 deg. A higher value is not recommended, to avoid that at the previous apogee the perigee is reduced to much. With an orbital period of nearly 6 days, the phase at the previous apogee is namely about 80 deg more. The small increase in apogee radius is obviously welcome.

TABLE 1. Effect of moon resonance: perigee x apogee radius = 30,000 x 250,000 km; inclination = 23 deg; argument of perigee w.r.t. moon orbit = 270 deg

phase w.r.t. moon (deg)	perigee radius increas (km)	apogee radius increas (km)	orbital plane rotation (deg)	line of apsides rotation (deg)
-31.60	1857	52	-0.87	-0.04
-25.07	2040	48	-1.26	0.09
-18.57	1999	32	-1.73	0.27
-12.09	1622	2	-2.21	0.44
-5.62	877	-41	-2.59	0.56
0.87	-92	-91	-2.73	0.58
7.42	-1006	-132	-2.55	0.50
14.03	-1626	-153	-2.14	0.36
20.68	-1887	-151	-1.63	0.19
27.36	-1862	-132	-1.16	0.04
34.03	-1658	-106	-0.78	-0.08

TABLE 2. Effect of moon resonance: perigee x apogee radius = 30,000 x 250,000 km; inclination = 31 deg; argument of perigee w.r.t. moon orbit = 180 deg

phase w.r.t. moon (deg)	perigee radius increas (km)	apogee radius increas (km)	orbital plane rotation (deg)	line of apsides rotation (deg)
-33.33	2171	586	-0.96	0.20
-26.76	2638	713	-1.16	0.46
-20.22	2940	805	-1.31	0.82
-13.74	2795	799	-1.31	1.24
-7.29	1867	604	-1.01	1.63
-0.85	194	168	-0.46	1.78
5.65	-1493	-366	0.67	1.61
12.24	-2478	-735	1.16	1.23
18.89	-2712	-849	1.31	0.83
25.55	-2500	-789	1.22	0.50
32.22	-2105	-657	1.02	0.24

At 200,000 km, the perigee radius is about 30,000 km which leads to an orbital period of nearly 4,5 days. With typical perigee thrust arcs between 240 and 280 deg in true anomaly the next moon resonance is reached in 5 revolutions. The desired phasing relative to the moon is achieved by tuning the size of the thrust arcs. In a similar way the subsequent moon resonances occur after respectively 4, 3, and 2 revolutions. Each time the change in perigee radius is increasing due to the closer distance to the moon.

Moon swing-by's

Moon resonances are encounters with the moon outside its sphere of influence. Once the distance of closest approach to the moon gets within the sphere of influence, i.e. lower than 60,000 km, we start speaking about moon swing-by's. The patched conics model provides a convenient way to qualitatively understand what happens at a moon swing-by.

During the period when the S/C is in the sphere of influence of the moon, it is on an hyperbolic trajectory w.r.t. the moon. The outgoing asymptotic velocity has the same magnitude as the incoming asymptotic velocity. The direction is changed in the plane containing the moon and the incoming asymptote. The deflection is towards the moon and increases with decreasing magnitude of the asymptotic velocity and decreasing distance between the moon and the incoming asymptote.

This allows to look at a swing-by as an instantaneous velocity change. The S/C trajectory prior to the swing-by is propagated ignoring the gravity of the moon until the point of closest approach is reached. At that point the velocity is changed according to the effect of the moon gravity. The trajectory is then further propagated ignoring the gravity of the moon.

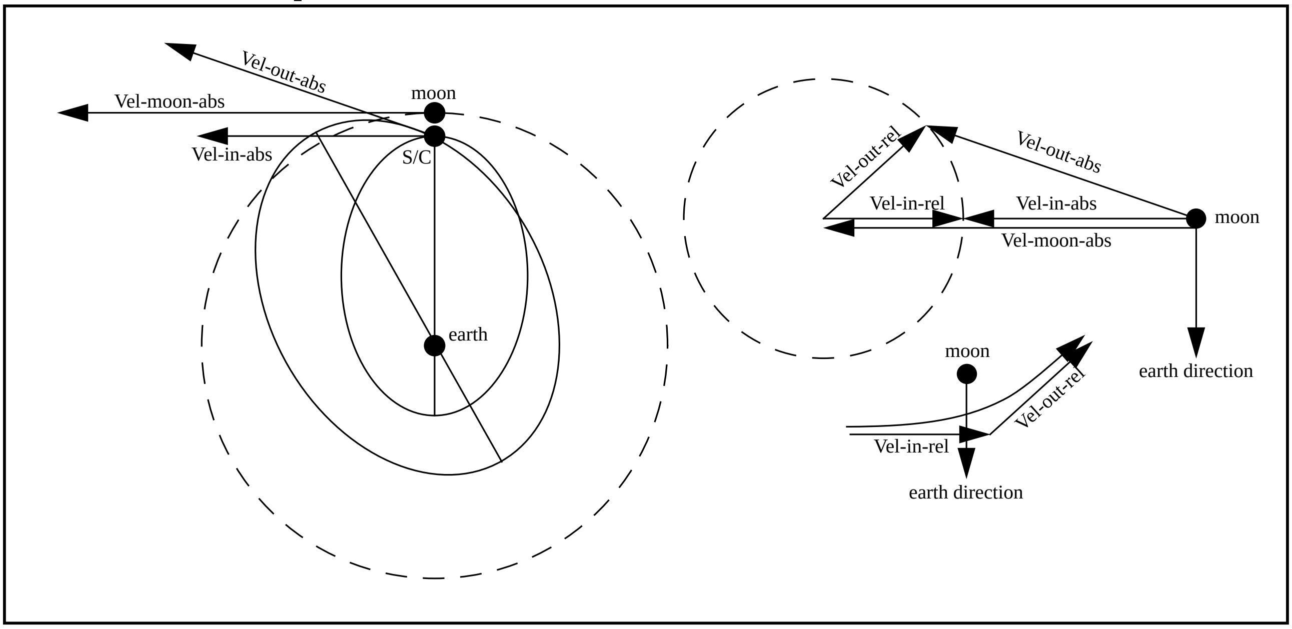
The incoming relative velocity is obtained as the difference between the S/C absolute velocity and the moon absolute velocity immediately prior to swing-by. This relative velocity is the asymptotic incoming velocity of the hyperbola w.r.t. the moon. It is rotated and added to the moon absolute velocity to get the S/C absolute velocity immediately after swing-by.

Figure 1 illustrates a swing-by at the apogee of an initial orbit which is eccentric and in the moon orbit plane. The incoming absolute velocity at apogee is along the moon velocity. Its magnitude is smaller because of the eccentricity. The incoming relative velocity is therefore opposite to the moon velocity. The apogee is slightly below the moon altitude, so that the point of closest approach is at the earth side of the moon. As a result, the relative velocity is rotated away from the earth direction. This leads to an absolute outgoing velocity with a positive radial and an increased tangential component. The orbit after the swing-by has a higher apogee and perigee radius. Also the line of apsides is rotated.

When in the case of Figure 1 the point of closest approach would be above the moon, the relative velocity, and hence the absolute velocity, would be bended downward. The inverse happens when the point of closest approach is below the moon. This provides a means to change the inclination to the moon orbit plane.

In the trajectories which we are considering, the swing-by's all occur close to the apogee of the initial orbit. With the apogee radius we control the change in perigee radius, apogee radius and the line of apsides and with the elevation of the apogee above the moon orbit we control the inclination to the moon orbit.

FIGURE 1.Example of a moon swing-by at the apogee of an initial orbit which is eccentric and in the moon orbit plane



Lunar Capture

The lunar capture trajectory and the subsequent transfer to the operational orbit around the moon is computed by backward integration. The operational orbit is polar and eccentric and has its perilune over the south pole. The perilune height is 1,000 km and the apolune height is 10,000 km. The right ascension of the ascending node in the moon equator as well as the arrival epoch in the operational orbit are free parameters.

Starting in the operational orbit, the trajectory is integrated backwards, applying a continuous thrust opposite to the velocity. The thrust spiral ends when the osculating apolune radius reaches 60,000 km. This provides the S/C with a large enough value of the Jacobian integral to leave the sphere of influence of the moon and consequently enter into the sphere of influence of the earth.

With the aim to get the S/C in an orbit with apogee below the moon orbit, the transfer to the earth sphere of influence is via the L1 point, which is the Lagrangian point of the earth-moon system between the earth and the moon, and not via the L2 point, which is at the other side of the moon. To avoid unstable trajectories, the transfer is direct, i.e. after thruster switch-off, the S/C leaves the sphere of influence of the moon without making an additional revolution around the moon.

For a given arrival epoch the desired transfer can only be achieved for a limited range of about 30 deg for the right ascension of the ascending node of the operational orbit. For changing arrival dates, the range shifts by roughly 360 deg per moon revolution. The resulting orbit within the sphere of influence of the earth has an apogee radius around 300,000, a perigee radius around 130,000 km and an inclination to the moon orbit plane around 12 deg. The apogee is almost at the ascending node in the moon orbit plane.

In general, these conditions can not be matched by a direct forward propagation from a GTO when apogee coast arcs are applied to reduce the fuel usage. The backward propagation is therefore extended by two pairs of revolutions in the sphere of influence of the earth. A first swing-by occurs near apogee after the first pair of revolutions. The point of closest approach is far below the moon at the earth side. This reduces the perigee and apogee radius and rotates the orbit plane by about 35 deg so that the apogee switches from the ascending to the descending node with a final inclination to the moon orbit around 23 deg. After the second pair of revolutions, a close moon swing-by is reached.

The parameters of the backward propagation are:

- the arrival mass, i.e. the mass at the start of the backward propagation
- the arrival epoch, i.e. the epoch at the start of the backward propagation
- the right ascension of the ascending node of the operational orbit in the moon equator
- the true anomaly range for the thrust arcs of the first pair of revolutions in the sphere of influence of the earth (opposite to the velocity)
- the true anomaly range for the thrust arcs of the second pair of revolutions in the sphere of influence of the earth (opposite to the velocity)

The arrival mass and the arrival epoch allow to control the mass and the epoch at the close moon swing-by. The right ascension of the ascending node and the sizes of the thrust arcs allow to control within limited ranges the magnitude of the relative velocity and the coordinates of the point of closest approach in earth direction and along the orbit normal.

Trajectory Matching

From the injection orbit the trajectory is propagated forward up to a close moon swing-by at which it is matched with the trajectory which is propagated backward from the operational orbit around the moon.

The relative velocity at the end of the forward propagation must be within the range achievable by the backward propagation (500 to 800 m/s). This allows to tune the parameters of the backward propagation to get the same relative velocity magnitude and to get a point of closest approach such that this relative velocity is rotated to the direction at the end of the forward propagation.

The sizes of the last and the last but one pair of perigee thrust arcs of the forward propagation are tuned to make the forward propagation ending at the same point of closest approach.

The matching of the mass and the epoch is achieved by tuning the mass and the epoch at the arrival in the operational orbit around the moon.

RESULTS

Ariane 5 injects the S/C in GTO, i.e. with a perigee radius of 6,962 km, an apogee radius of 42,164 km, an inclination (INC) of 7 deg and an argument of perigee (AOP) of 178 deg. The right ascension of the ascending node (RAAN) depends on the day of launch and on the lift-off time. As additional passenger, we must be prepared to accept all values for the RAAN from 0 to 360 deg. After a few revolutions in GTO, a continuous thrust spiral along the velocity is started until the perigee radius reaches 20,000 km. The aim is to stop the degradation of the solar arrays as soon as possible. During this phase the apogee radius is increased to 68,000 km. Due to the earth oblateness, the RAAN regresses by about 10 deg and the AOP increases by about 20 deg.

Subsequently, the apogee radius is further increased using thrust arcs symmetric around perigee with the aim to reach a close moon swing-by with a relative velocity within the range achievable by the backward propagation. The nature of this trajectory is strongly affected by the orientation of the S/C orbit w.r.t the moon orbit plane when reaching the perigee radius of 20,000 km. Table 3 shows this orientation for the whole range of possible values of the RAAN.

TABLE 3. Typical orientation of S/C orbit w.r.t. moon orbit when reaching perigee radius of 20,000 km.

RAAN in ME2000 system (deg)	Inclination to moon orbit (deg)	Arg. of perigee to moon orbit (deg)	Elevation of apogee to moon orbit (deg)
0	19.6	2.7	-0.9
30	20.1	42.4	-13.4
60	22.6	78.8	-22.1
90	26.1	110.2	-24.4
120	29.6	137.9	-19.3
150	32.2	163.4	-8.8
180	33.4	187.9	4.3
210	33.2	212.2	17.0
240	31.4	237.0	25.9
270	28.5	263.1	28.3
300	24.9	291.8	23.0
330	21.6	324.8	12.3
360	19.6	2.7	-0.9

Favorable Case

A trajectory has been generated for a launch on 21 September. The RAAN when reaching the perigee radius of 20,000 km is 147 deg. This can be considered as a favorable case because the point on the orbit crossing the moon orbit plane is close to apogee. The apogee is raised with the proper phasing by tangential thrust arcs symmetric around perigee until a close swing-by with the moon is reached. Moon resonances are used to already increase the perigee radius prior to the swing-by. The point of closest approach at the moon swing-by is controlled by tuning the size of the thrust arcs between the last resonance and the swing-by and the size of the thrust arcs between the last but one and the last resonance.

The backward propagation from the operational orbit around the moon back to the close moon swing-by is as explained earlier. The relative velocity w.r.t. the moon at the end of the forward propagation is within the range of possible relative velocities at the end of the backward propagation so that both trajectories can easily be matched at the swing-by.

The transfer extends over 464 days and involves 3 moon resonances and 2 moon swing-by’s. It requires 54.3 kg of fuel which corresponds to a velocity increment of 2,67 km/s.

Unfavorable Case

More problematic is a launch on 6 February which leads to a RAAN of 282 deg. The resulting argument of perigee of 271 deg, when reaching the perigee radius of 20,000 km, combined with an inclination of 26 deg w.r.t. the moon orbit makes the apogee to be 26 deg above the moon orbit plane. It is not sufficient anymore to raise the apogee radius to the moon altitude to get a close moon swing-by. To nevertheless achieve this in a direct way, one could increase the apogee radius far above the moon altitude until the point of 90 deg or 270 deg true anomaly reaches the moon. Another possibility would be to reduce the inclination w.r.t. the moon orbit to zero by applying an out-of-plane thrust direction modulation around the points of 90 deg and 270 deg true anomaly. Also possible is a rotation of the line of apsides by 90 deg or -90 deg using in-plane thrust direction modulation or thrust arcs asymmetric around perigee. All these techniques significantly increase the fuel needed to reach the moon. An additional moon swing-by at a relatively large distance allows to partially reduce this fuel penalty. This leads to a total transfer duration of 484 days and a fuel usage of 59.9 kg which corresponds to a velocity increment of 2,95 km/s.

Moon resonances rotate the orbit plane around the line of apsides, so that the argument of perigee increases above 270 deg at the expense of an increased inclination. An even more difficult case seems therefore to occur for a launch on 29 December which leads to a RAAN of 246 deg. The argument of perigee w.r.t. the moon orbit increases from initially 240 deg to 255 deg after the last resonance. The resulting trajectory for this case is shown in Figure 2, 3, 4 and 5. The effect of the resonances and the swing-by’s on the S/C orbit is shown in Table 4. The parameters of the swing-by’s are presented in table 5 and table 6. The transfer duration is 517 days an requires 60.9 kg of fuel which corresponds to a velocity increment of 3.03 km/s

For both cases, the apogee radius is increased to the moon altitude by direction modulated thrust arcs symmetric around perigee, simultaneously increasing the semi-major axis and decreasing the elevation of the apogee above the moon orbit. The ratio between the two weighting factors is selected to reduce the apogee elevation to 13 deg after the last resonance.

The reduction of the apogee elevation is accomplished by out-of-plane thrust direction modulation to rotate the orbit plane and in-plane thrust direction modulation to rotate the line of apsides. Moon resonances are used to increase the perigee radius. The reduction of the apogee elevation to 13 deg allows the S/C to just enter the sphere of influence of the moon when the apogee is at the moon altitude. The phasing is controlled to have this far swing-by 10 deg prior to apogee. The S/C is then lagging behind the moon. The major result is a significant increase in perigee radius and a rotation of the orbit plane roughly around the line from the earth to the point of closest approach. Also the apogee radius is slightly increased. This brings the point on the orbit which is located 10 deg after apogee inside the moon orbit plane near the moon altitude. The post swing-by period is selected to get a close moon swing-by at this point at the occasion of the next moon passage. The arrival conditions for the second moon swing-by are controlled by tuning the size of the thrust arcs between the last resonance and the first swing-by and the size of the thrust arcs between the first and the second swing-by. The relative velocity at the second swing-by is within the range of possible relative velocities at the end of the backward propagation so that both trajectories can easily be matched at the swing-by.

FIGURE 2.Launch on 29 Dec.(triangle base down = thrust start; triangle base up = thrust end; circle = apogee; square = swing-by; box = start/end)

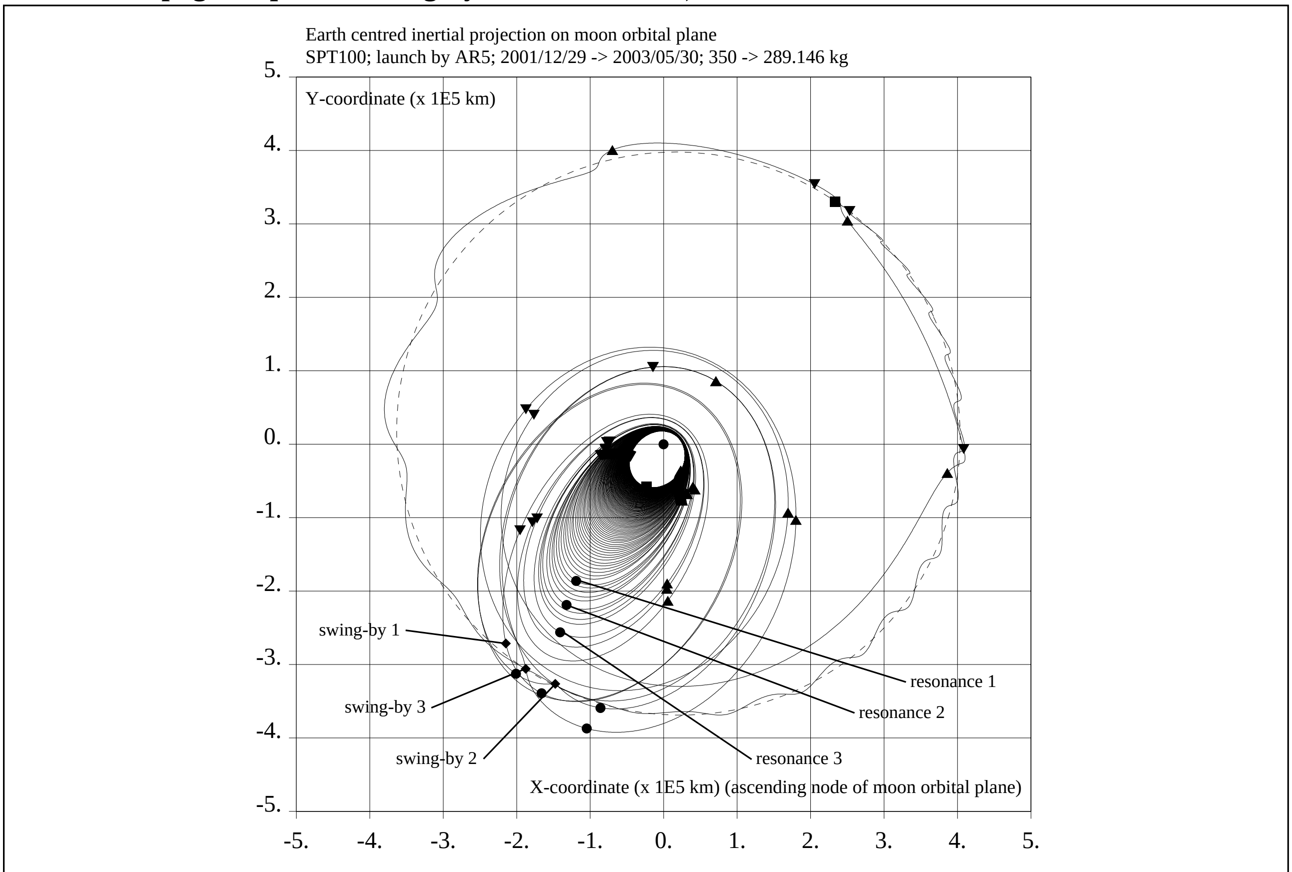


FIGURE 3.Launch on 29 Dec.(triangle base down = thrust start; triangle base up = thrust end; circle = apogee; square = swing-by; box = start/end)

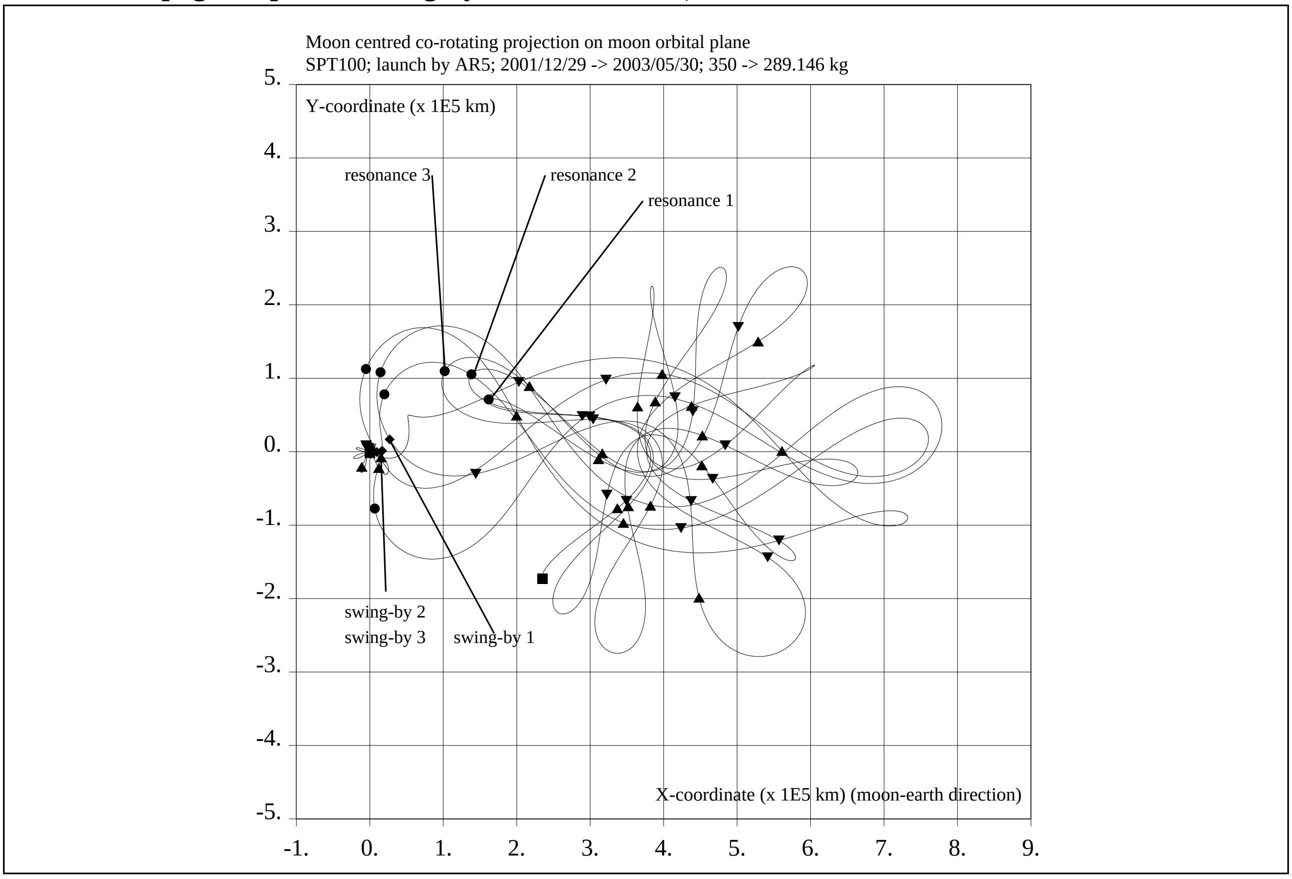


FIGURE 4.Launch on 29 Dec.(triangle base down = thrust start; triangle base up = thrust end; circle = moon; box = start/end)

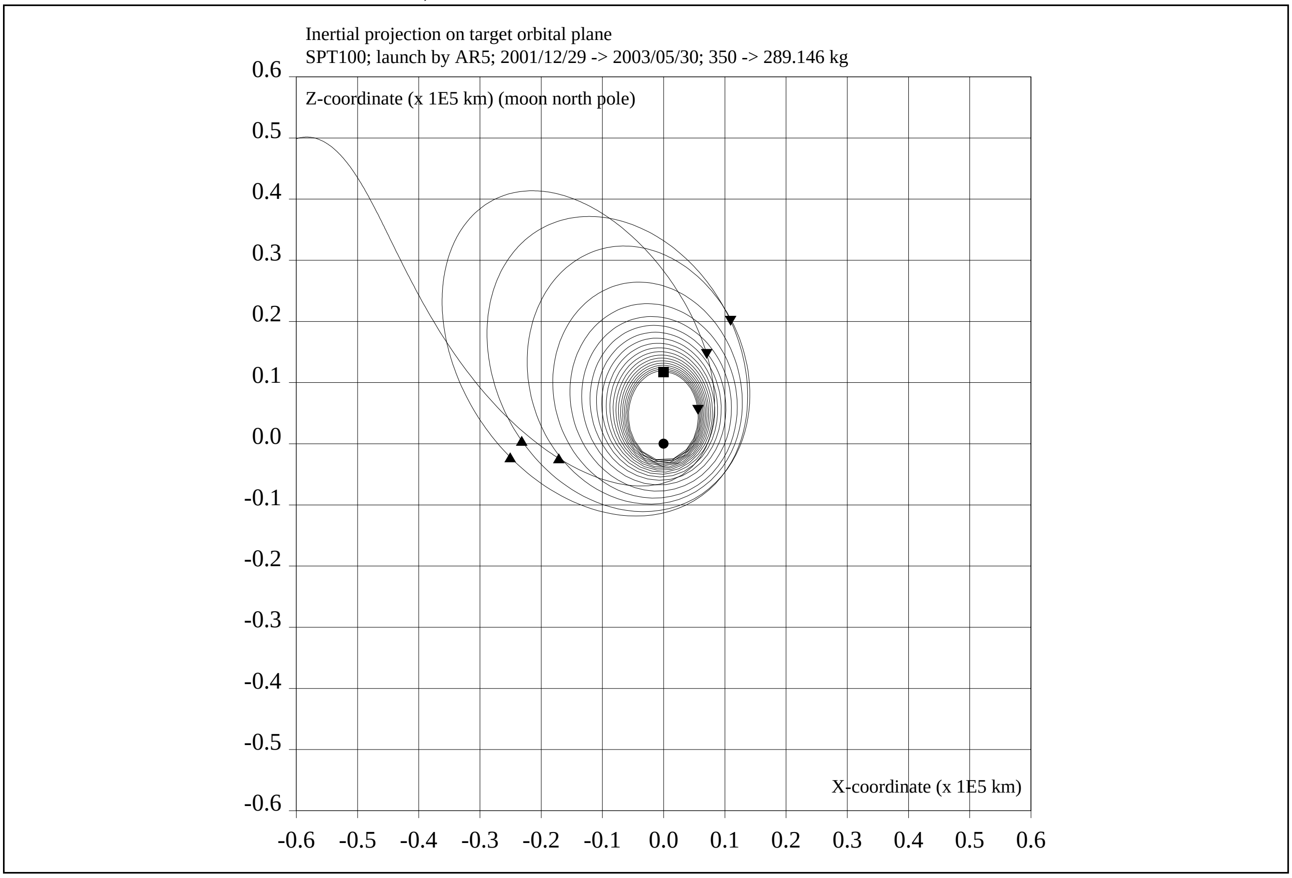


FIGURE 5.Launch on 29 Dec.(triangle base down = thrust start; triangle base up = thrust end; circle = apogee; square = swing-by; box = start/end)

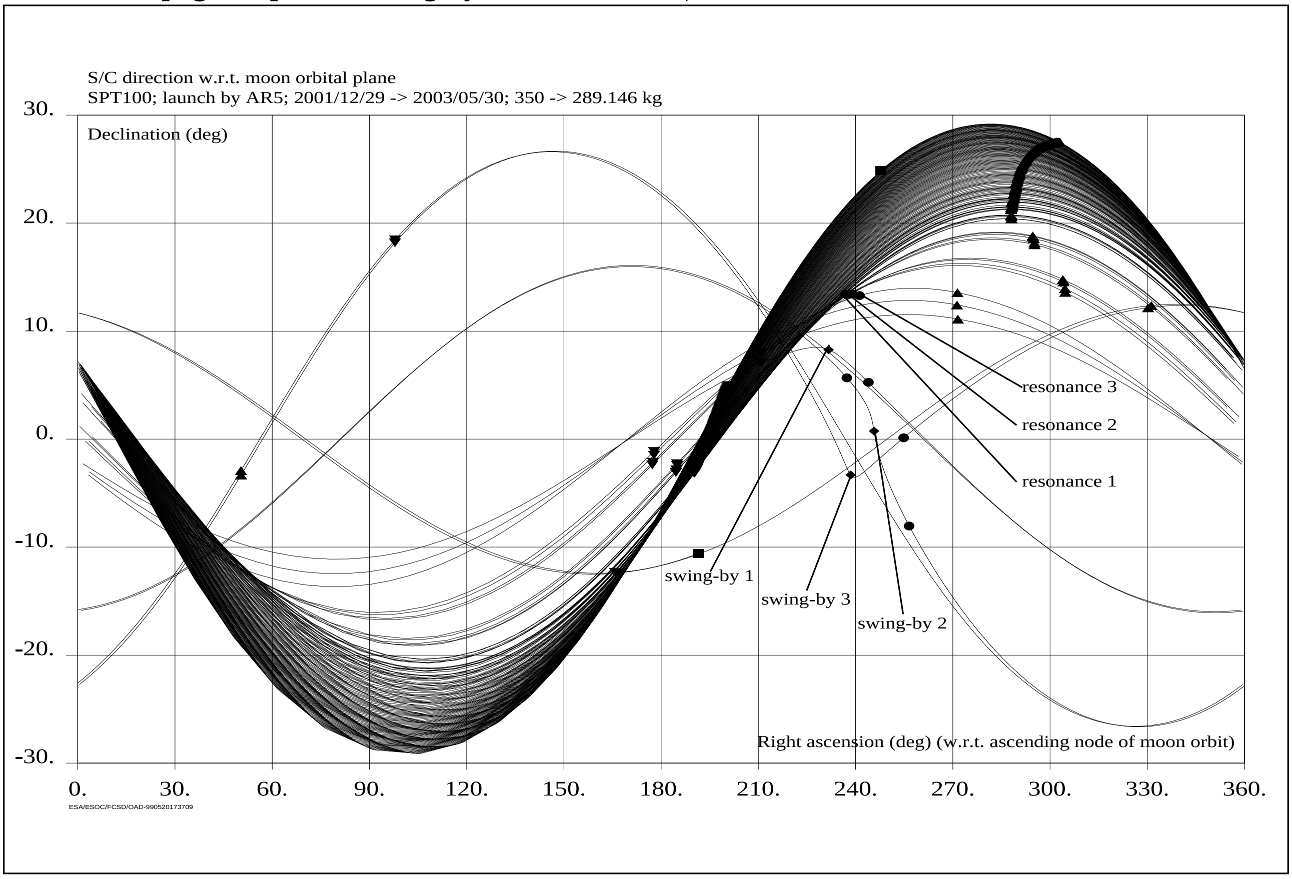


TABLE 4.Launch on 29 Dec.: orbital elements before and after resonances, swing-by's and capture.

		perigee radius (km)	apogee radius (km)	INC w.r.t. moon orbit (deg)	AOP w.r.t. moon orbit (deg)
resonance 1	before	21,894	225,534	20.3	221.8
	after	22,955	226,316	19.1	225.6
resonance 2	before	22,340	260,195	18.4	227.7
	after	25,130	262,125	16.7	234.7
resonance 3	before	26,191	294,955	16.0	238.8
	after	32,718	300,283	13.9	254.3
swing-by 1	before	37,321	361,469	10.0	254.7
	after	74,394	383,435	16.1	339.0
swing-by 2	before	76,075	380,688	15.9	339.6
	after	107,895	373,605	26.6	18.2
swing-by 3	before	107,710	351,136	26.6	19.2
	after	130,140	412,981	12.3	176.9
capture	before	125,679	300,094	12.5	182.2

TABLE 5.Launch by on 29 Dec.: asymptotic velocities w.r.t. the moon at swing-by's

		to earth (m/s)	opposite to moon velocity (m/s)	normal to moon orbit (m/s)	norm (m/s)
swing-by 1	before	-255	531	92	596
	after	-357	467	-96	596
swing-by 2	before	286	433	-163	544
	after	-220	392	-306	544
swing-by 3	before	-79	320	-297	444
	after	-271	329	123	444

TABLE 6.Launch on 29 Dec.: points of closest approach to the moon at swing-by's.

	to earth (km)	opposite to moon velocity (km)	normal to moon orbit (km)	distance to moon (km)
swing-by 1	2,715	16,861	50,353	59,623
swing-by 2	16,844	1,338	4,730	17,547
swing-by 3	9,455	-435	-20,784	22,838

LONG ECLIPSE AVOIDANCE

Giving a midnight launch in GTO, earth eclipse periods occur shortly after launch and repeat each 6 months during transfer. Eclipses due to the moon occur in moon orbit. Eclipses due to the earth also occur in moon orbit around the equinoxes. Typical eclipses for some transfer trajectories are shown in Table 7.

TABLE 7.Eclipses during transfer as function of launch date (U=Umbra, U+P=Umbra+Pre-umbra).

Launch Date	Total		> 0.5 hours		> 1 hour		> 1.5 hours		> 2 hours		Max. Dur. (h)	
	U	U+P	U	U+P	U	U+P	U	U+P	U	U+P	U	U+P
2002/02/02	141	143	73	76	0	0	0	0	0	0	0.85	1.00
2002/04/02	121	125	47	58	6	8	6	7	5	6	3.19	3.72
2002/06/15	154	155	87	94	23	27	0	0	0	0	1.21	1.25
2002/09/15	98	99	5	5	4	5	4	5	3	5	2.97	3.46

The battery capacity requires the eclipse umbra to be limited to 2 hours. The acceptable duration of the pre-umbra depends on the fraction by which the sun is occulted. The following classifies the eclipses which can occur and identifies the possibilities to reduce their duration to an acceptable value.

After Launch

A large number of short eclipses occur near perigee. Due to the large velocity they are all shorter than 2 hours.

6 months after launch

A cluster of up to 14 eclipses with the longest about in the middle occurs near apogee. The relative low velocity near apogee may cause several of these eclipses to be longer than 2 hours. They account for the majority of the long eclipses in Table 7. The worst case occurs when the apogee lies in the ecliptic plane, i.e. for a launch near the equinoxes.

The duration of these eclipses can be reduced below 2 hours by moving the apogee away from the ecliptic plane prior to the eclipse period. After the eclipse period, the apogee has to be moved to the moon orbital plane to get a close encounter with the moon. Unfortunately, the moon orbital plane is close to the ecliptic plane. This change away from and back to the ecliptic plane is quite expensive in terms of additional fuel usage (up to 6.5 kg for launch in spring or autumn).

Another way to reduce the duration of these eclipses to a value around 2 hours is to introduce a waiting period (no thrust) between the end of the solar array degradation phase (perigee radius = 20,000 km; 2.5 months after launch) and the eclipse period. The effect is a reduced apogee radius during the eclipse period. This will increase the number of eclipses because of the shorter orbit period and closer distance to the earth. The longest eclipse duration is reduced because of the larger apogee velocity. In the worst case a waiting period of 4.5 months and an additional fuel usage of 1.5 kg may be required.

12 months after launch

A few short eclipses occur near perigee. Due to the large velocity they are all shorter than 2 hours.

18 months after launch

The apogee altitude is already near the moon altitude also in case of an increased transfer duration to reduce the duration of the eclipses 6 months after launch. One gets at most 2 sometimes long eclipses (up to 9 hours total duration). They can be reduced by changing the phase of the S/C in the orbit at a cost of less than 1 kg of fuel.

In moon orbit

For the target 10,000 x 1,000 km polar orbit, eclipses by the moon are always shorter than 2 hours.

In spring and autumn, when the sun is close to the Moon orbit plane, an eclipse by the earth with a total duration up to 7 hours may occur. The umbra is limited to 3.5 hours. The total duration can usually be reduced below 2 hours by changing the phasing of the S/C in its moon orbit which requires less than 0.5 kg additional fuel. In one particular case (eclipse on 24 April 2005) it is required to aim for a suboptimal right ascension of ascending node for the orbit around the moon which can lead to a substantial fuel penalty. One should however note that the major part of the eclipses by the earth have a large pre-umbra portion.

NAVIGATION AT MOON APPROACH

The orbit determination uncertainty on the approaches to the moon for swing-by's and insertion is expected to be lower than 100 km in the B-plane. The B-plane is the plane perpendicular to the asymptotic relative velocity at approach. The real error in the B-plane might be larger, because the S/C might not be able to correct for a known error in the orbit due to the limited thrust level available. As a result, the post swing-by orbits might have considerable errors which have to be corrected by modifying the post swing-by transfer trajectory. The associated fuel cost as function of the error in the B-plane is shown in Figure 6, Figure 7 and Figure 8 for the moon swing-by's when launching on 29 December 2002. The arrow indicates the direction to the moon centre. As can be seen even a 1,000 km B-plane error can be corrected with less than 1 Kg. Also the moon insertion has been analyzed in a similar way and proven not to be very critical as regards pure navigation errors.

Far more dangerous is a non-nominal performance of the thrust arcs prior to the lunar swing-by's and insertion. If for instance the last thrust arc prior to lunar insertion is aborted prematurely, the S/C will impact on the moon or perform a swing-by leading to an earth escape trajectory.

FIGURE 6.Launch on 29 Dec.: Fuel penalty to correct navigation errors at 1st lunar swing-by.

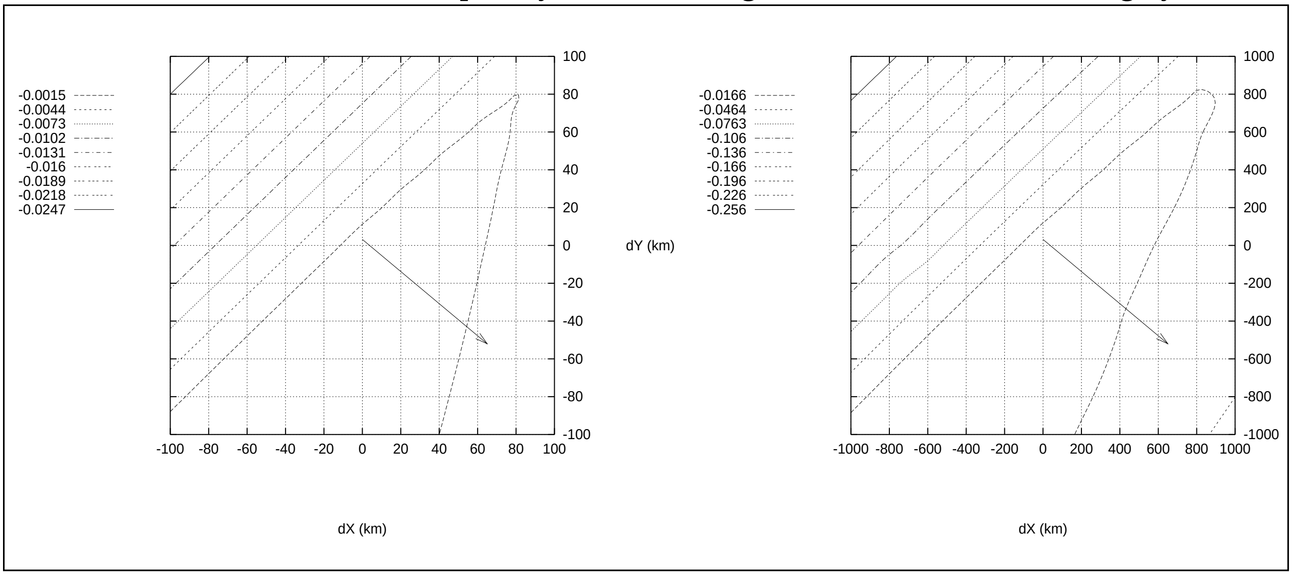


FIGURE 7.Launch on 29 Dec.: Fuel penalty to correct navigation errors at 2nd lunar swing-by.

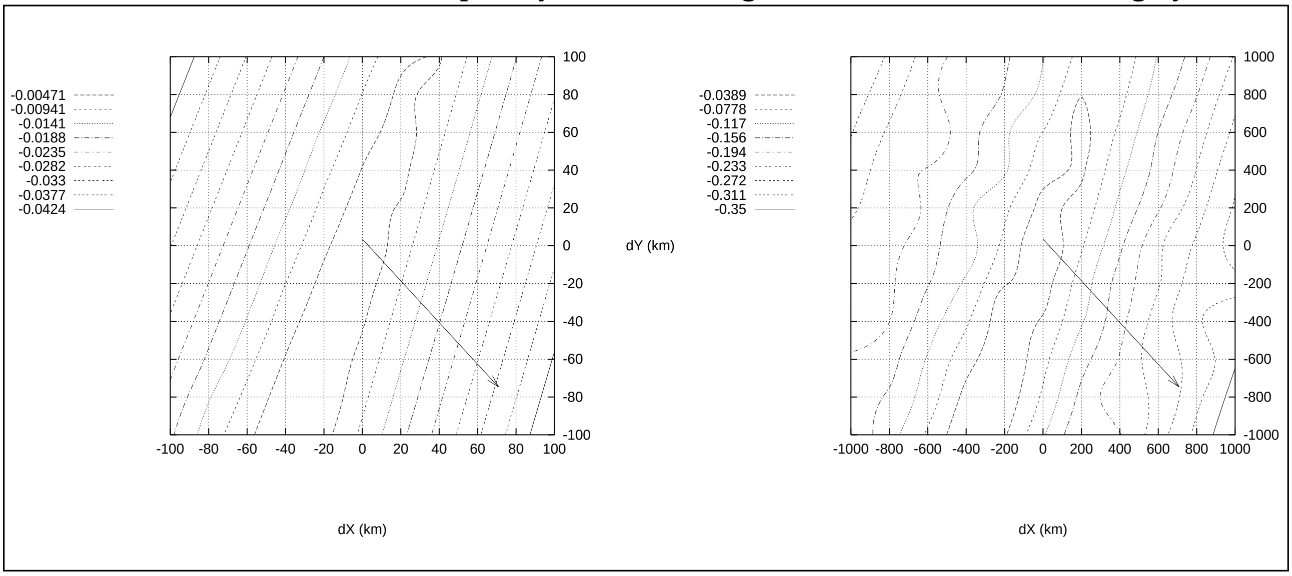
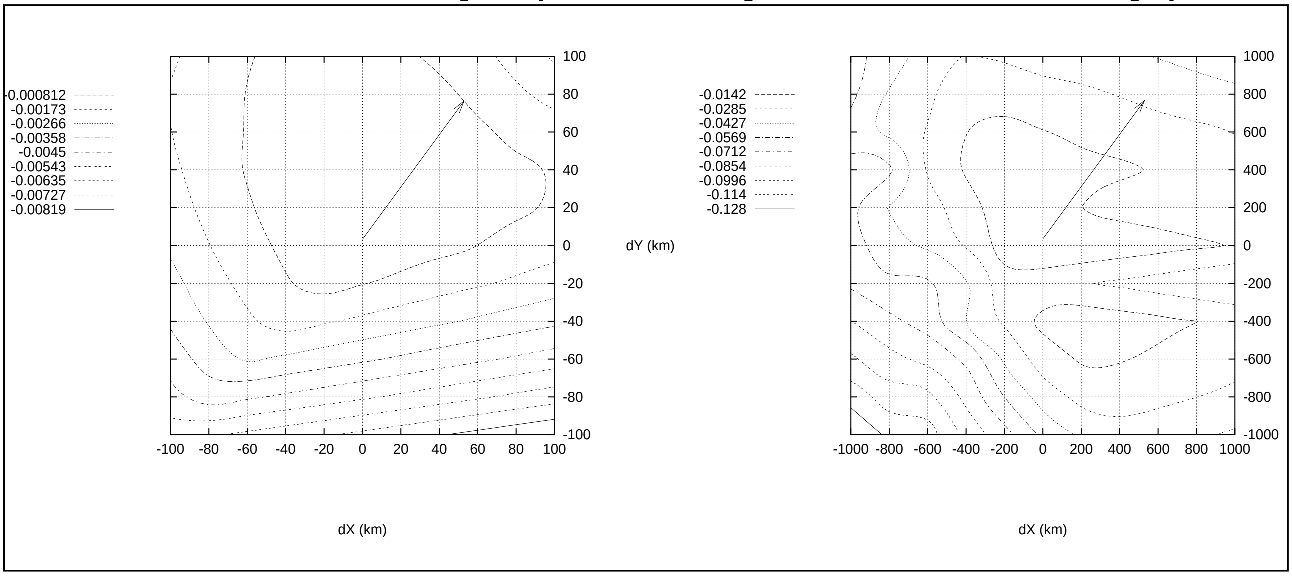


FIGURE 8.Launch on 29 Dec.: Fuel penalty to correct navigation errors at 3rd lunar swing-by.



CONCLUSION

Using a pragmatic engineering approach without applying optimization techniques, fuel efficient transfer trajectories have been constructed for the SMART-1 moon mission, by combining low thrust and multiple moon swing-by's. The transfer duration is about 17 months. The fuel demand for a 350 kg launch mass ranges from 54.3 kg to 60.9 kg depending on the launch date.

The ascent part of the mission, i.e. up to the first resonance, has been further optimized using averaging techniques combined with the Pontryarchin principle (ref. 2 and ref. 3) showing a potential for fuel reduction up to 2 kg. This has been confirmed by applying a direct optimization technique (ref. 1) to the entire transfer, resulting in a benefit between 2 and 3 kg.

For launches in spring and autumn special measures have to be taken to limit the eclipse duration to 2 hours. These measures will either increase the fuel consumption by up to 7 kg or increase the transfer duration by 4.5 months.

The transfer strategy is robust against navigation errors of typically 100 km on the approaches to the moon prior to swing-by's and insertion. More problematic is a premature termination of the last thrust arcs prior to these events.

REFERENCES

1. SMART-1 Mission Analysis: Trajectory design using the moon gravity; May 1999; S1-ESC-RP-5501; J. Schoenmaekers, J. Pulido, J. L. Cano (ESOC)

2. Generalisation des techniques de moyennation en controle optimal: Application aux problemes de transfer et rendez-vous orbitaux a poussee faible; S. Geffroy (Institut National Polytechnique de Toulouse); 1997

3. Optimum Low Thrust Transfer Between Two Orbits; R. Jehn and J. L. Cano (ESOC); MAS Working Paper 414; March 1999

4. SMART-1: Consolidated Report om Mission Analysis, Issue 1.2, J. Cano, M. Hechler, D. Horas, M. Khan, J. PUlido, J. Schoenmaekers (ESOC), July 2001, S1-ESC-RP-5506